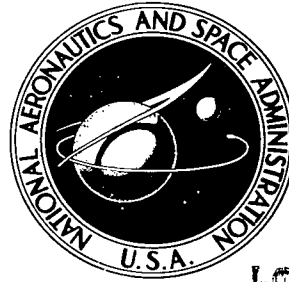


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NASA TN D-8314 c.1

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**A FLIGHT INSTRUMENTATION SYSTEM
FOR ACQUISITION OF ATMOSPHERIC
TURBULENCE DATA**

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Charles W. Meissner, Jr.
December 1976

Page 13: The units of the pressures in table I are incorrect. Replace table I with the attached corrected version.

Issued January 1978



0134014

1. Report No. NASA TN D-8314		2. Government Accession No.		3. Recipient's Catalog No.	
4. Title and Subtitle A FLIGHT INSTRUMENTATION SYSTEM FOR ACQUISITION OF ATMOSPHERIC TURBULENCE DATA				5. Report Date December 1976	
				6. Performing Organization Code	
7. Author(s) Charles W. Meissner, Jr.				8. Performing Organization Report No. L-10596	
				10. Work Unit No. 505-07-31-02	
9. Performing Organization Name and Address NASA Langley Research Center Hampton, VA 23665				11. Contract or Grant No.	
				13. Type of Report and Period Covered Technical Note	
12. Sponsoring Agency Name and Address National Aeronautics and Space Administration Washington, DC 20546				14. Sponsoring Agency Code	
15. Supplementary Notes					
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17. Key Words (Suggested by Author(s)) Flight instrumentation Instrumentation Atmospheric turbulence			18. Distribution Statement Unclassified - Unlimited Subject Category 06		
19. Security Classif. (of this report) Unclassified	20. Security Classif. (of this page) Unclassified	21. No. of Pages 20	22. Price* \$3.25		

A FLIGHT INSTRUMENTATION SYSTEM FOR ACQUISITION OF ATMOSPHERIC TURBULENCE DATA

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SUMMARY

A flight instrumentation system for the acquisition of atmospheric turbulence data is described. Airflow direction transducers and an impact-pressure transducer are the primary instruments for measuring vertical and lateral gust velocity, and a sensitive incremental pressure transducer is used to measure longitudinal gust velocity. Airplane motions, sensed by an inertial platform, are subtracted from the primary measurements during postflight data reduction to yield true gust-velocity time histories. Salient engineering features of the instrumentation are discussed, and a complete description of the instrumentation is presented.

INTRODUCTION

Treatment of aircraft gust-loads response as a random process phenomenon requires a clear definition of the power spectrum of atmospheric turbulence. (See refs. 1 and 2.) Increased flight activity at supersonic speeds and the desire to verify theoretical models of the atmosphere have intensified interest in turbulence characteristics at the longer wavelengths of the power spectrum where it is not well defined. Historically, the long-wavelength portion of the spectrum has been difficult to measure because instrumentation drift is excessive when the time histories are required to be tens of minutes in duration. A sampling program has been designed which uses the technique of Crane and Chilton (ref. 3) to derive atmospheric turbulence power spectra from in-flight turbulence encounters. The technique has been improved in that an inertial platform is used as the velocity reference to obtain greater accuracy at the longer wavelengths and in that lateral and longitudinal components of gust velocity are measured. Also, modern data recording and processing techniques allow larger statistical samples to be processed. A B-57B airplane has been instrumented at Langley Research Center to collect in-flight turbulence data. It is the purpose of this report to describe the onboard instrumentation system used to collect the turbulence data.

The concept of the sampling program is to fly the airplane through a turbulent air mass, which produces an airflow about the airplane as a result of the turbulence of the air mass and the velocity of the airplane. Airflow measurements are made in the undisturbed air forward of the airplane by using an instrumented boom extended from the nose. Three components of airflow fluctuations are measured along the lateral, longitudinal, and normal axes of the airplane. Measurements are also made of the airplane attitude and linear velocity relative to the Earth by means of an inertial platform located in the airplane, and of

attitude rate. The instrumentation measures and records time histories of these quantities and total air temperature.

All measurements are conditioned, sampled, digitized, and recorded on one track of an airborne tape recorder in a serial data stream. Time and pilot comments are also recorded on other tape tracks. The digital tape is subsequently edited by the experimenter and selected portions are submitted for spectral analysis. Airflow attributable to airplane motion is subtracted from the total airflow data during postflight data reduction. The result is the time history (for each component) of airflow in the turbulent air mass during the encounter. The final result of the data reduction is a spectral presentation of the gust velocities.

SYMBOLS

a_x	acceleration in inertial system x-direction
a_y	acceleration in inertial system y-direction
g	acceleration due to gravity ($1g = 9.8 \text{ m/sec}^2$)
l	distance from inertial platform to flow vanes
p	static pressure
p_t	total pressure
q_c	impact pressure
T	total temperature
V	reference volume
V_{ax}	change in velocity in inertial system x-direction during the record period
V_{ay}	change in velocity in inertial system y-direction during the record period
V_{az}	change in velocity in inertial system z-direction during the record period
V_T	true airspeed, m/sec
V_x	velocity in inertial system x-direction
V_y	velocity in inertial system y-direction
w_g	vertical gust component

x,y	inertial system horizontal coordinates (The inertial system horizontal coordinates are orthogonal and bear a known but complicated relationship to geographic coordinates. For the range of distances covered by the turbulence sampling flights, x is approximately east and y is approximately north.)
z	inertial system vertical coordinate
α	relative angle of attack
β	relative angle of sideslip
Δa_n	incremental normal acceleration at aircraft center of gravity (1g bias)
Δa_z	incremental acceleration in inertial system z-direction (1g bias)
Δp	incremental static pressure
Δp_t	incremental total pressure
Δq_c	incremental impact pressure
$\Delta \psi$	incremental azimuth angle
θ	pitch attitude
$\dot{\theta}$	pitch rate
ϕ	roll attitude
$\dot{\phi}$	roll rate
ψ	heading angle
$\dot{\psi}$	yaw rate

Abbreviations:

AIDS	airborne inertial data system
GSE	ground support equipment
PCM	pulse code modulation
RC	resistance capacitance
rms	root mean square

METHOD

Background and details of the method used to combine measurements can be found in references 1 and 2, but the form of the gust velocity determination is illustrated by the following equation for vertical gust component w_g :

$$w_g = V_T(\alpha - \beta - \theta) + V_{az} + \dot{h} \quad (1)$$

The first term on the right-hand side of equation (1) represents the vertical component of airflow relative to the airframe, as sensed by the flow vanes where small angle approximations have been made. The vertical speed of the vanes relative to the inertial system is given by the last term. The remaining term (V_{az}) relates the measurements to the Earth. This equation results in cancellation of the apparent gust velocity due to airplane motion relative to the air mass (except for a constant). Since the data are in the form of time histories of the individual airflow and motion measurements, these must correlate in time so that the differences between motion-induced and total airflow can be computed at the same instant. Airplane forward velocity is much greater than the gust velocity of the turbulent air mass, which means that the actual gust angles (w_g/V_T) are small and the motion correction terms in the equation are relatively large. An inherent difficulty with this measurement technique is that the data result from taking the small difference between large quantities. This problem is especially pronounced at an aerodynamically resonant frequency such as the Dutch-roll frequency.

SYSTEM DESCRIPTION

Instrumentation Installation

A photograph of the airplane and some of the instrumentation is shown in figure 1. Balsa flow vanes, used for vertical and lateral airflow measurements, are mounted on the nose boom behind the static-pressure ports. The total-pressure port, used for longitudinal airflow measurements, is located in the nose of the boom.

Two statoscopes are mounted in the nose compartment. One statoscope is used to measure incremental static pressure, and the other is used to measure incremental total pressure. A static-pressure transducer, an impact-pressure transducer, a resistance bridge, and amplifiers are mounted in a heated box in the nose compartment. The outputs of all the pressure transducers and the resistance bridge are amplified to fill the ± 5 -volt input range of the data system. The temperature probe extends into the airstream from the side of the nose compartment. It is a platinum thermometer element the resistance of which is measured by the resistance bridge.

The pilot instrumentation controls and displays are in the front cockpit, as shown in figure 2. The instrumentation controls are in a control panel located above the pilot instrument panel. From left to right, there is a failure warning light located over a record light. The failure warning is derived from the inertial platform malfunction indicator. A tape-remaining meter indicates the amount of data tape available for recording. A readout of the instru-

mentation time code generator provides correlation of pilot-observed events with recorded measurements. This correlation can be made on the voice track of the data tape. An instrumentation power switch and a record switch are located to the right of the time readout.

To assist the pilot in evaluating the turbulence level a turbulence intensity meter is provided on the instrument panel. This meter displays the root-mean-square (rms) value of the output of a normal-axis accelerometer. The rms output is averaged over a pilot-selectable period of from 2 to 20 sec and has a range of 0 to 1g.

The positions of the angle-of-attack and angle-of-sideslip vanes and airplane pitch attitude are available in one combined pilot display. A separate display is used for incremental heading. These are sensitive measurements and it is easily within the pilot's capability to cause the measurement outputs to exceed the ± 5 -volt range of the data system and thus lose the data. Part of the piloting task is to keep these indicators on scale during turbulence encounters.

Located in the rear cockpit and not shown in figure 2 is a monitor for the temperature probe output; this is provided for the use of the in-flight meteorological observer. Also, a remote indicator for the turbulence intensity meter is located in the rear cockpit. The rate gyros and the airborne inertial data system (AIDS) are located under the rear cockpit seat. The AIDS is a production inertial navigation system which has been modified for this experiment.

A normal accelerometer is mounted at the top of the bomb bay to measure acceleration at the center of gravity. This output is not used in the equations for airflow measurement but is used as an aid in editing the raw data before processing begins and can also be used for airplane response studies.

The instrumentation pallet is mounted on the bomb-bay door in an insulated box (shown with cover removed in fig. 3). The rotatable bomb-bay door is at the 90° position for access to the instrumentation. Normal preflight procedure is to turn on and check out the instrumentation, install the pallet cover, and rotate the bomb bay to its closed position. From that time through engine start and completion of the flight, the instrumentation remains on. Instrumentation turn-on is performed in the hangar. A battery cart powers the instrumentation inverter and is moved with the airplane as it is towed onto the ramp. After engine start when the airplane generators are supplying power, the instrumentation inverter is switched to the airplane generators and the battery cart is removed. When postflight instrumentation check-out is required, the procedure is reversed. This preflight procedure is used since the AIDS must be aligned with the Earth axis of rotation. The most accurate alinement is obtained in the hangar where winds cannot disturb the airplane during the alinement period. If power is removed after alinement, the process must be repeated. The instrumentation inverter and power control unit are located in a compartment aft of the bomb bay.

Ground support equipment is housed in two relay racks. This equipment is used to calibrate and verify the performance of the data system and the pressure transducers. The AIDS has ground support equipment to verify its performance by exercising factory-provided tests which must fall within prescribed limits.

Airflow Measurement Transducers

Figure 4 shows the airflow measurement transducers. The transducers are located both in the nose compartment and on the boom. At the top of the figure is shown a platinum resistance thermometer which measures total temperature. It is operated into a resistance bridge and is calibrated over a range of -50°C to 50°C . The statoscope setup used for measuring incremental total pressure is shown in the lower part of the figure. This instrument is normally used to detect small changes in pressure altitude but is being applied here to the measurement of longitudinal airspeed fluctuations. The instrument utilizes a differential pressure transducer with the input and reference sides of the diaphragm interconnected through a valve. The valve is normally open to equilibrate pressure. When a measurement is required, the valve is closed and the pressure increment is indicated by using a sensitive pressure transducer. Small pressure fluctuations can be accurately measured even though the total range of input pressure is greater than the range of the transducer. The reference volume V is a constant-temperature reference volume of high thermal mass which is used to minimize the pressure changes in the transducer reference and associated tubing.

The statoscope is housed in a heated box, and is kept at a constant temperature of 37.8°C to provide a stable pressure reference and a favorable environment for the transducer. Although only one transducer is shown in the diagram, two transducers, differing in sensitivity, are used. The more sensitive transducer is required at high altitudes where pressure fluctuations are small, and the less sensitive transducer stays on scale at the lower altitudes. An identical statoscope is used to measure static-pressure changes.

For minor pilot corrections which increase pitch attitude, the airplane gains altitude and slows. Both of these effects tend to reduce the total pressure and drive the output of the incremental total-pressure transducer out of range of the data recording system. In order to relieve this problem, incremental static pressure is electrically subtracted from the incremental total pressure to result in incremental impact pressure Δq_c which is the recorded measurement.

The incremental total-pressure transducer operates from the total-pressure port on the front of the boom. The tubing and volumes associated with the total-pressure system exhibit pneumatic resonances and the fundamental resonance affects the high-frequency end of the data band. A constriction is used at the total-pressure port to flatten the response of the system and make it uniform over the data band. This constriction is changed depending on the anticipated flight altitude.

The static-pressure line provides the reference pressure for the impact-pressure transducer and is connected to the static-pressure transducer and the incremental static-pressure statoscope.

The angle-of-attack and angle-of-sideslip vanes are located on the boom and are attached to synchros. The converter demodulates the synchro signal to a voltage proportional to vane position. The outputs of the vane system, the pressure transducers, and the temperature transducer are presented as voltage analogs with a range of -5 volts to 5 volts.

Airplane Response Measurements

The airborne inertial data system and three gyros measure the response of the airplane during turbulence encounters. The rate gyros are mounted to indicate angular rates about the three airplane axes. The AIDS is a special adaptation of a 15-year-old inertial navigation system which is no longer in production in the United States. Some of the salient features of the present 7-year-old system are indicated in figure 5. The inertial platform is gimballed and has three accelerometers and two gyros mounted on it. The accelerometer and gyro outputs are used as inputs to an analog computer which drives the platform gimbals and gyros through torque motors to keep the platform aligned with the local vertical and oriented to the inertial-system reference direction. The output circuits electronically integrate the platform-mounted accelerometer outputs. The integrators are directly coupled to provide response to the lowest frequencies of interest. Integration is started at the beginning of a record when the initial condition of zero is removed by opening the relay across the feedback capacitor. The integrator then provides an incremental velocity output for the duration of the record. The platform attitude angles are measured by synchros attached to the platform gimbals. The synchro outputs are demodulated by synchro-to-dc converters which are matched to those used to demodulate the flow vane position. This is done to minimize the differential phase shift between these measurements. Since, for this experiment, the airplane operates over an angular range from -7.5° to 7.5° in pitch and from -20° to 20° in roll but operates over 360° in azimuth, the azimuth resolution is coarse by comparison. In order to increase azimuth resolution, a track-and-hold amplifier was provided to be used as a reference. With the record off, the amplifier nulls the incremental azimuth $\Delta\psi$ output to zero. When the record is initiated, the amplifier holds the last azimuth value as a reference by opening the relay on the track-and-hold amplifier. The difference between azimuth output and the reference value is amplified to the same sensitivity as the pitch output.

Data System

All the instrument outputs are voltage analogs of the measured quantities at the input to the data system. Figure 6 is a block diagram of the data system. The inputs enter the data system through antialiasing filters. Passive, single-pole, RC filters with a 3-dB frequency of 20 Hz are built into the system. Four-pole, Butterworth response, active filters are optionally available for high attenuation above 32 Hz.

The filters have been matched to the quantities being measured to obtain negligible phase shift between the measurements which are to be combined in any one equation during the data reduction process. The passive filters have an attenuation of 1 dB at 10 Hz. This attenuation is easily calculated and may be combined with other correction factors when the final power spectrum is calculated.

The pulse code modulation (PCM) system samples and encodes each input signal with a 10-mV resolution. The PCM system has 80 channels, of which 25 are used for turbulence measurements. The remaining channels are used for synchronization and engineering measurements or are filled with a code to aid in signal

synchronization during data tape playback. The PCM system output is a 160 kilobit, Miller code data stream. These data are recorded directly on one track of a flight tape recorder. Time code and pilot comments are recorded on two other tracks of the recorder. The tape recorder provides 90 minutes of record time at 0.38 m/sec tape speed.

Figure 3 shows the data system as mounted on the pallet with the components identified. Also identified are the power supplies, ground support equipment connector, flow vane, and inertial system synchro-to-dc converters.

Instrumentation System Accuracies and Sources of Error

Table I is a listing of the instrumentation system measurements, the range of each measurement, and the estimated errors derived from calibrations. Both static and dynamic accuracies are included. Where quantities vary with flight conditions, values are given for an altitude of 12.19 km and a true airspeed of 192 m/sec. To calibrate the angle measurement instrumentation, direct calibrations using precision dividing heads were taken through the system when it was mounted on the airplane. Pressure measurement instrumentation was laboratory calibrated and periodically checked for calibration change by using high-quality pressure transducers. Measurement variability resulting from the periodic checks is combined with the calibration errors by the root-sum-square method to yield a total error estimate. Power-supply voltages and instrument ambient temperatures during flight are kept to the values used for instrument calibration which allowed supply-voltage and temperature variation effects to be neglected. Both power-supply voltages and instrument temperatures are monitored during flight.

Total-temperature, acceleration, and roll-, pitch-, and yaw-rate overall static errors were estimated by taking the root sum square of the voltage calibration error with a ± 2 -count equivalent error from the PCM system.

Systems dynamics were measured in the laboratory to obtain constants (cut-off frequency, or natural frequency, and damping) for the first- or second-order approximation of the dynamic characteristics of the instrumentation used to make each measurement. Dynamic characteristics of the flow vanes and the incremental impact-pressure pneumatic system vary with altitude and airspeed. These effects have been extrapolated from sea-level or wind-tunnel measurements.

Relative phase-shift errors as shown in table I were derived from the same measurements and calculations as were used to determine amplitude response, with the exception of the platform attitudes θ , ψ , ϕ , and $\Delta\psi$. An estimate of the total phase shift for platform attitudes was obtained by comparing direct and through-the-platform zero crossing time delays measured during tilt-table movements. These tests showed that the platform time constants add 0.75 phase shift to the synchro-to-dc converter phase shift at 1 Hz. Pitch angle is the reference for the phase measurements. The phase-shift differences decrease linearly to zero at zero frequency.

Because of the need to determine very low frequency and phase responses, a procedure for determining these responses involved the use of the flight PCM

system and a general-purpose computer as shown in figure 7. The input and output of the unit under test were sampled by the PCM system and stored in the computer in real time. The stored samples were then plotted and printed, and the amplitude and phase differences between input and output were computed from the printout.

Total Temperature

The total-temperature-probe amplitude response appears to be quite low. This is misleading because the response of the probe is composed of two time constants and the probe indicates 70 percent of true reading with a frequency response in excess of 0.5 Hz.

Angles of Attack and Sideslip

Response of the instrumentation for the angle-of-attack measurement is considered in two parts. The first part is the aerodynamic response of the vane and synchro and the second part is the response of the demodulator and data system. The aerodynamic responses of the balsa flow vane and synchro have been investigated in wind-tunnel tests by Richardson and are presented in reference 4. The results of reference 4 were used to extrapolate corrections for aerodynamic response which are combined with corrections for data-system dynamic response and can be applied to the power spectrum during data reduction if required. The second part of the response is from the synchro shaft through the data system. This response has been measured directly by sinusoidally rotating the synchro shaft at various frequencies and observing the output of the PCM system.

A static error can result from insufficient aerodynamic lift to overcome bearing friction. Wind-tunnel tests (ref. 4) and bearing friction measurement of several vanes led to calculations which show that the anticipated limit of resolution for these flow vanes lies between 0.02° and 0.09° with the aerodynamic restoring force available at 5.6 mPa because of sticking and friction in the connecting shaft bearings. Tests indicate that this remains constant to -56.57°C as long as lubricant is very sparingly applied to the bearings. Total measurement response varies with altitude, but the dynamic amplitude response of the total system is more uniform than the synchro and data system since the aerodynamic response (second-order, underdamped) of the flow vane compensates for the response of the synchro and data system (first-order). This causes the lowest system 1-dB bandwidth (shown in table I) to occur where the vanes taken alone have the most uniform response, since the roll-off of the PCM system is uncompensated. The phase error of the flow vane and incremental impact pressure change with altitude. Estimates of system errors are given for a 12.19-km altitude and 192-m/sec true airspeed. These errors do not include the errors caused by flow direction changes due to upwash effect which are carried by the flow field of the airplane.

Incremental Total Pressure

Frequency response for the incremental total-pressure measurement is adjusted for altitude by selecting a predetermined constriction for the pitot head. Selection of the constriction and calculation of the resulting frequency response were accomplished by extrapolating sea-level measurements to flight altitude by using the second-order model provided by reference 5. Frequency response values are shown for a 12 192-m altitude and 192-m/sec true airspeed.

One constriction is used for altitudes below 9144 m. The resulting response, calculated from the second-order model, is underdamped at sea level, flat at 4572 m, and overdamped at 9144 m. Above 9144 m the full tubing area is used as the inlet. Damping increases with altitude and at about 13 716 m the system has its best uniformity. The low resonant frequency and flat response give the incremental impact pressure measurement the highest relative phase shift of any of the measurements in table I.

The second-order model that was used to describe the total-pressure system is a simplification of a more nearly exact analysis. It was selected to provide a basis for predicting the effects of density and viscosity variations with altitude and airspeed. The plumbing associated with the pitot port is 3.97 m of 6.35-mm tubing with a tap located 3.13 m from the pitot port. Transducer volume is negligible. A 0.53-m length of 6.35-mm tubing is connected to the tap. It is terminated by a length of 3.18-mm tubing and the q_c transducer. A theoretical analysis of the tubing network was not attempted. Instead, sea-level measurements were used to arrive at what might be termed a natural or resonant frequency and damping for the pneumatic system. The effort was directed toward arriving at constrictions which would make the response curve reasonably flat over the operating range. The resulting calculations, based on the second-order model, predicted a ± 1 dB bandwidth of from 5 to 11 Hz, depending on altitude.

AIDS

The remaining measurements in table I are derived from the AIDS. AIDS accuracy depends on the accuracy of its three accelerometers, the stability of its three gyros, and the system mechanization. The accelerometers are factory calibrated since they are integral parts of the AIDS platform and are tested to an rms accuracy specification of 0.0001g. This accuracy is 20 times better than data system resolution, so the accelerometers are considered to contribute no error to the measurements of acceleration. The steady error is nulled out of the incremental velocity integrators by using the bias set shown in figure 5. A postflight check for zero velocity after each flight gives a measure of how well the accelerometers and other parts of the AIDS are performing. An order-of-magnitude change in accelerometer characteristics would appear as a significant deterioration in postflight performance. Gyro drift is periodically checked and nulled as required. Gyro drift specification (0.01 deg/hr) is 10 times better than data system resolution over a 10-min data period, so this error source is considered to be zero.

The AIDS mechanization results in error buildup in the system which increases in time because unbounded integrations are used in the control loops.

The gyros are also sensitive to the lateral accelerations which occur during turns. These errors are not discernible by the data system in the acceleration or attitude outputs, but are readily apparent in the velocity outputs. Once the AIDS is in error, the error appears as a sine wave with a period of 84.4 min, called the Schuler period. The amplitude of this sine wave on the velocity outputs is recorded after each flight and is used as a measure of AIDS performance during the flight. Postflight velocity errors range from 0.61 m/sec to 6.1 m/sec peak to peak. This error source is not included in table I since it appears as a discrete frequency in the final spectrum determined from the data. These errors give the AIDS-derived velocities a low frequency of 0.0002 Hz, below which the estimated errors are much higher than the stated values.

Platform attitude outputs were calibrated through the data system with the platform mounted on a tilt table. The estimated errors given in table I are for nonlinearity and hysteresis. The incremental velocities are obtained by electronically integrating the accelerometer outputs. AIDS ground-support equipment checks the accuracy of integration by integrating a test signal. There is no specification for total V_x and V_y and there is no calibration since an adequate test range is not available. The measured error in this case is the root sum square of single point checks of the maximum velocity error at the end of each flight. Since inertial-system errors tend to increase with flight time, this error measure averaged over many flights gives an upper value of platform velocity errors.

In an 18-month period, a total of 46 flights were made with the instrumentation system at altitudes ranging from sea level to 14 630 m. For a series of 25 flights the average postflight Schuler V_x and V_y errors were 4.8 m/sec and 3.4 m/sec peak to peak, respectively. This amount of Schuler oscillation could result in a maximum trend error of 1.75 m/sec and 1.24 m/sec in the resulting horizontal components of gust velocity if a 10-min record was taken at the zero crossing point of the sine wave and if the platform heading corresponded to the x- or y-direction of measurement and calculation. The AIDS time constants which significantly affect the dynamic characteristics in the frequency range from 0 to 2 Hz are associated with the electronics in the output signal unit. The AIDS stable platform has three major sources of dynamic response errors. These are the accelerometers, the platform-leveling servo system, and the platform shock mounts. The accelerometer natural frequency is about 75 Hz, so its effect on the measurement dynamics was disregarded. The servo bandwidth lies between 40 and 50 Hz. Its effect was disregarded for amplitude response, but a small phase shift was found. The platform shock mount system has a natural frequency of 23 Hz, but the damping ratio is so low (on the order of 0.02) that its effect on response was disregarded. Dynamic responses of the accelerations and incremental velocities are controlled by capacitors in the feedback path of the output amplifiers. The attitude responses are controlled by the responses of the synchro-to-dc converters. These responses were measured by applying an electrical or mechanical sine-wave stimulus as appropriate and recording the result through the data system.

SYSTEM EXPERIENCE AND RECOMMENDATION

In an 18-month period, a total of 46 flights were made with the instrumentation system at altitudes ranging from sea level to 14 630 m. Results in general were satisfactory. The efforts expended in controlling temperature and supply voltage variations resulted in a system with low random noise without expending the effort to validate system performance over the wide variation in environmental conditions experienced by the airplane. The primary source of error and unreliability was the inertial system. This 7-year-old system, which is a special adaptation of a 15-year-old design, is no longer in production in the United States. Thus, routine support was difficult to obtain. The most significant improvement in the instrumentation system would be the substitution of a modern inertial system with better performance, fewer mechanical components, and readily available support. Another desirable feature would be an inertial system which could be operated by the flight crew. This feature would permit operations from remote airfields and increase the chances for turbulence encounters and thus increase the efficiency of the instrumented airplane.

Langley Research Center
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December 2, 1976

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TABLE I.- INSTRUMENTATION SYSTEM MEASUREMENTS

Measurement	Range	Estimated error, 1 σ	± 1 dB frequency response, Hz	Relative phase shift at 1 Hz, deg
α	-7.5° to 7.5°	^a 0.09°	^b 26 ^b 10	-0.75
β	-7.5° to 7.5°	^a .09°	^b 26 ^b 10	-.75
$\dot{\phi}$	-1 to 1 rad/sec	.01 rad/sec	11	-.75
$\dot{\theta}$	-1 to 1 rad/sec	.008 rad/sec	10	-.75
$\dot{\psi}$	-0.5 to 0.5 rad/sec	.005 rad/sec	10	-.75
T	-50° to 50° C	.3° C	.5	
^c q _c	0 to 34.5 kPa	.166 kPa		
^c p	0 to 103.4 kPa	.476 kPa		
^c Δp	-1.723 to 1.723 kPa	^a .020 kPa		
	-0.689 to 0.689 kPa	^a .006 kPa		
^c Δq_c	-1.723 to 1.723 kPa	^a .048 kPa	11	-6.50
	-0.689 to 0.689 kPa	^a .014 kPa	11	-6.50
Δa_n	-1g to 1g	.008g	10	2
a_x	-1g to 1g	.005g	7	-1.5
a_y	-1g to 1g	.005g	7	-1.5
$\Delta \ddot{a}_z$	-3g to 3g	.013g	7	-1.5
V_{ax}	-30.48 to 30.48 m/sec	.122 m/sec	7	-1.5
V_{ay}	-30.48 to 30.48 m/sec	.122 m/sec	7	-1.5
V_{az}	-30.48 to 30.48 m/sec	.122 m/sec	7	-1.5
ϕ	-20° to 20°	^a .08°	5	0
θ	-7.5° to 7.5°	^a .03°	10	Reference
$\Delta \psi$	-7.5° to 7.5°	^a .04°	10	0
V_x	0 to 304.8 m/sec	1.75 m/sec		
V_y	0 to 304.8 m/sec	1.24 m/sec		
ψ	0° to 360°	.7°	10	0

^aNonlinearity and hysteresis.^bLowest system bandwidth.^cDifferential pressure.

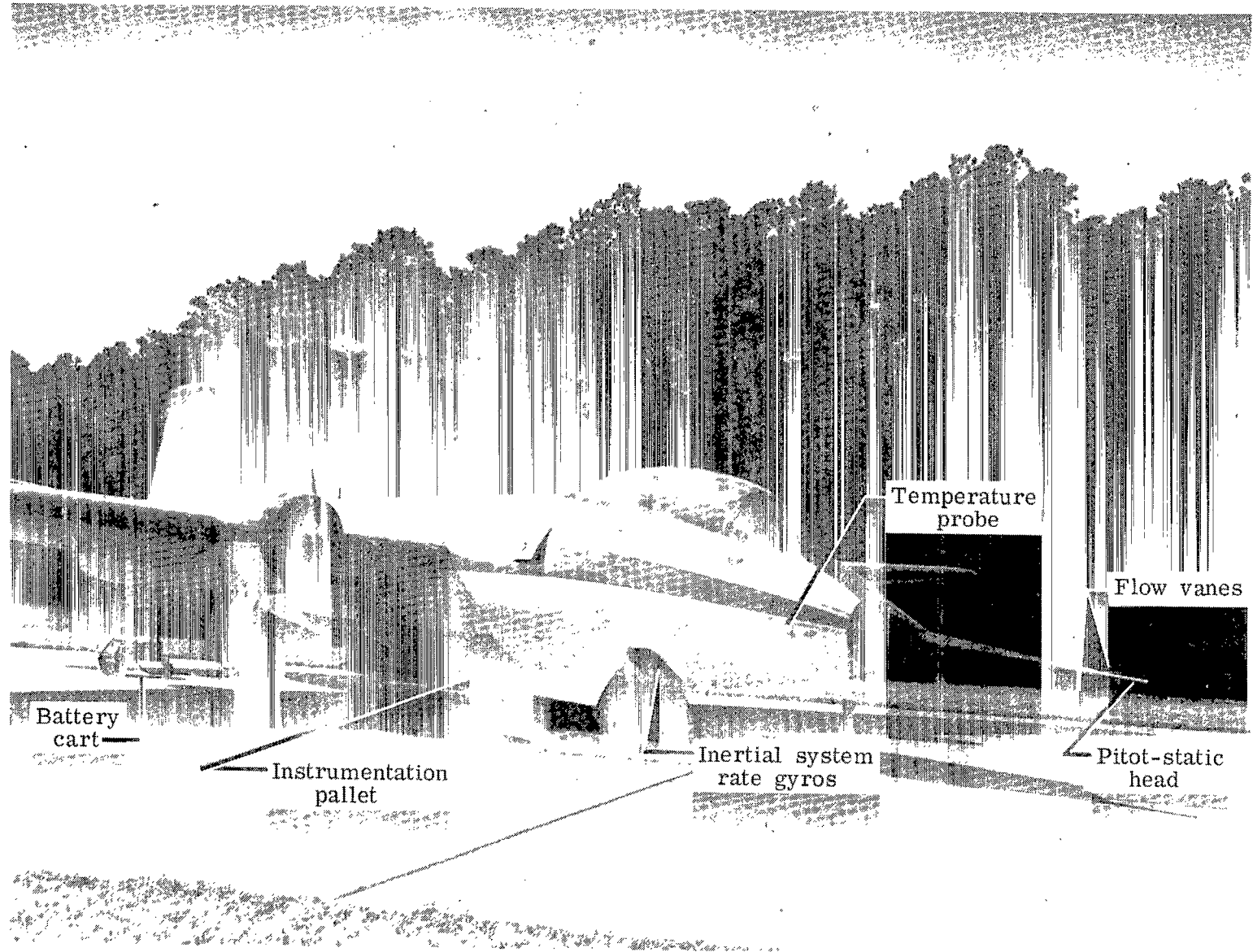


Figure 1.- Instrumented airplane.

L-76-7512

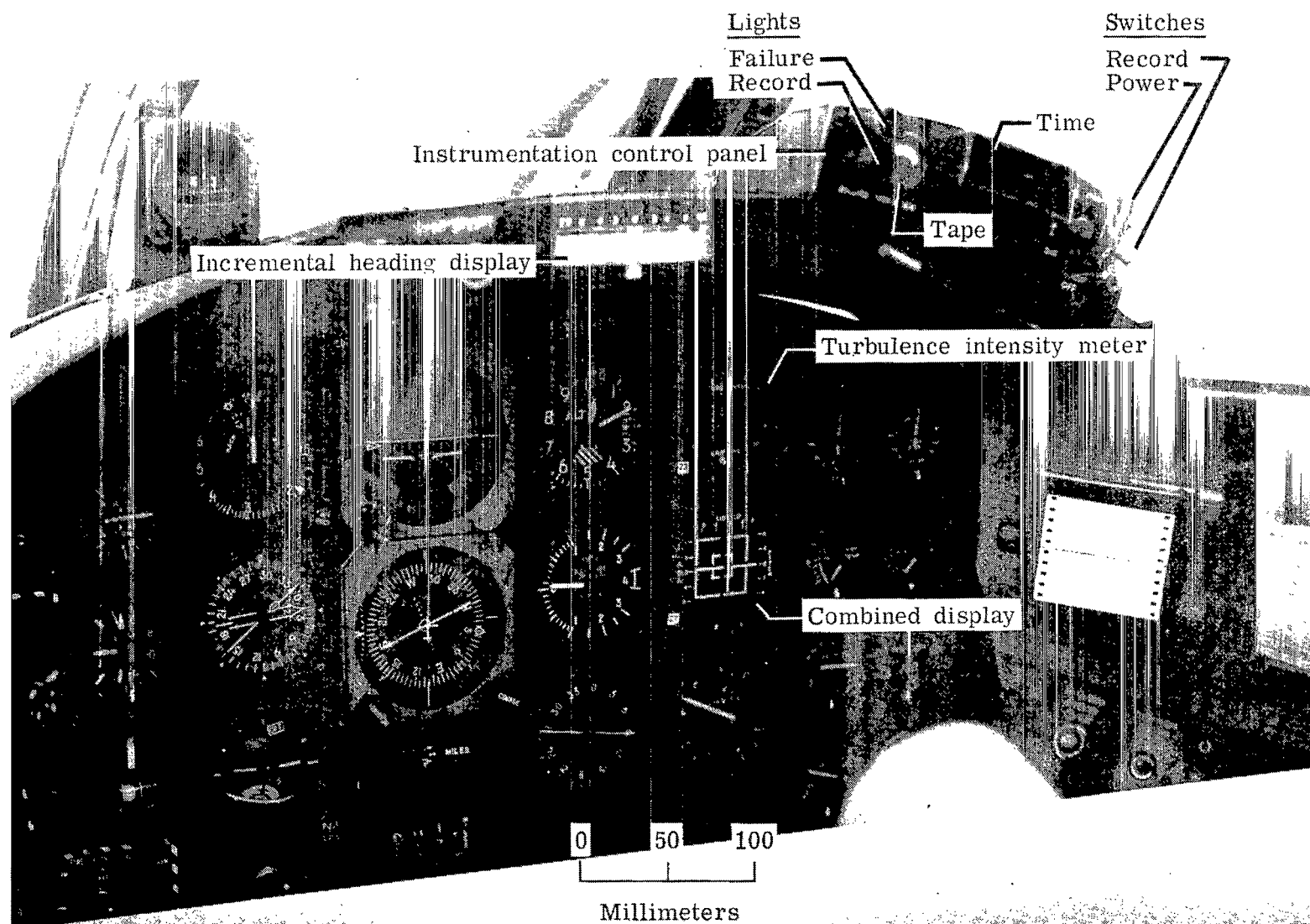


Figure 2.- Instrumentation controls and displays.

L-76-1405.1

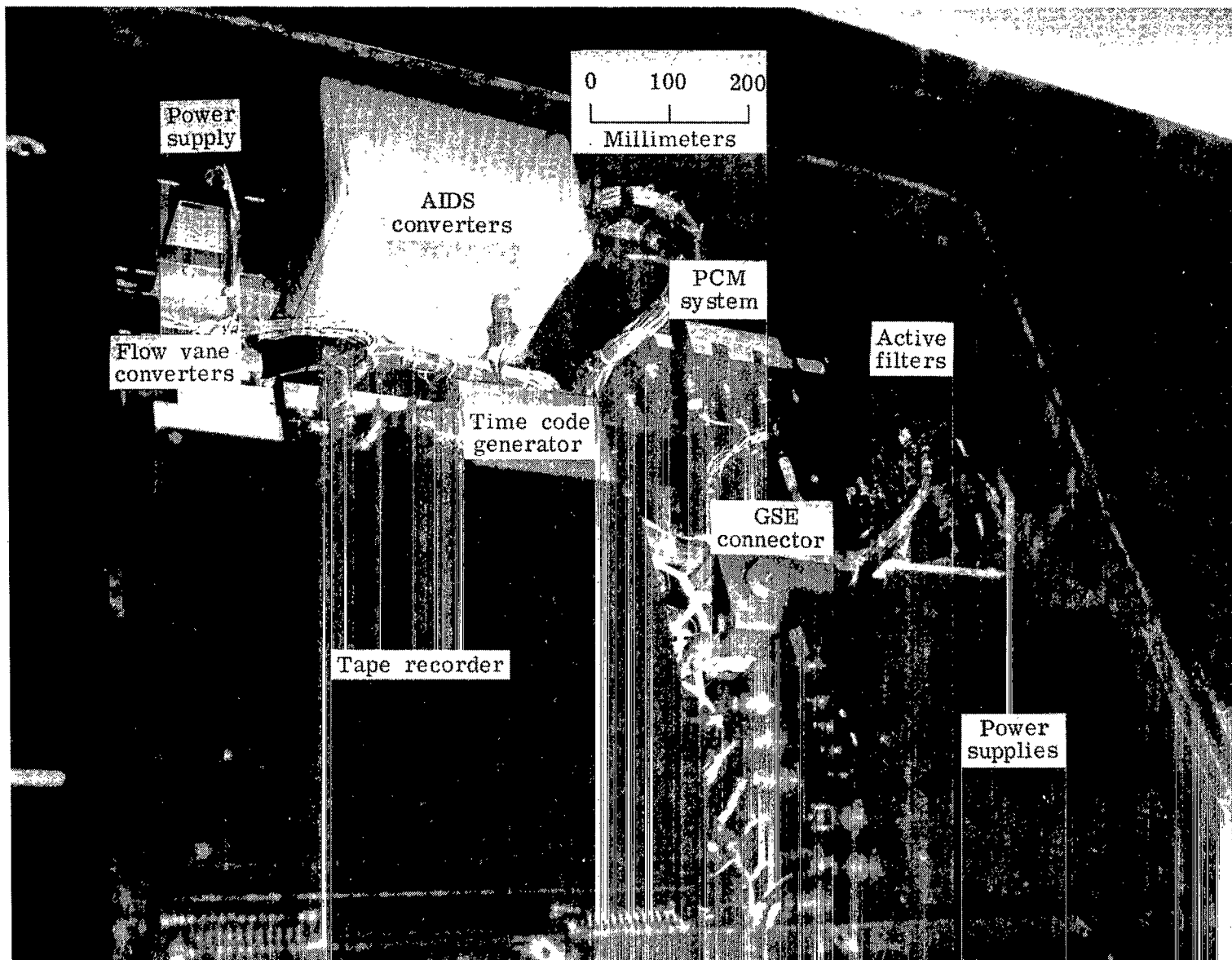


Figure 3.- Instrumentation pallet.

L-75-7560.1

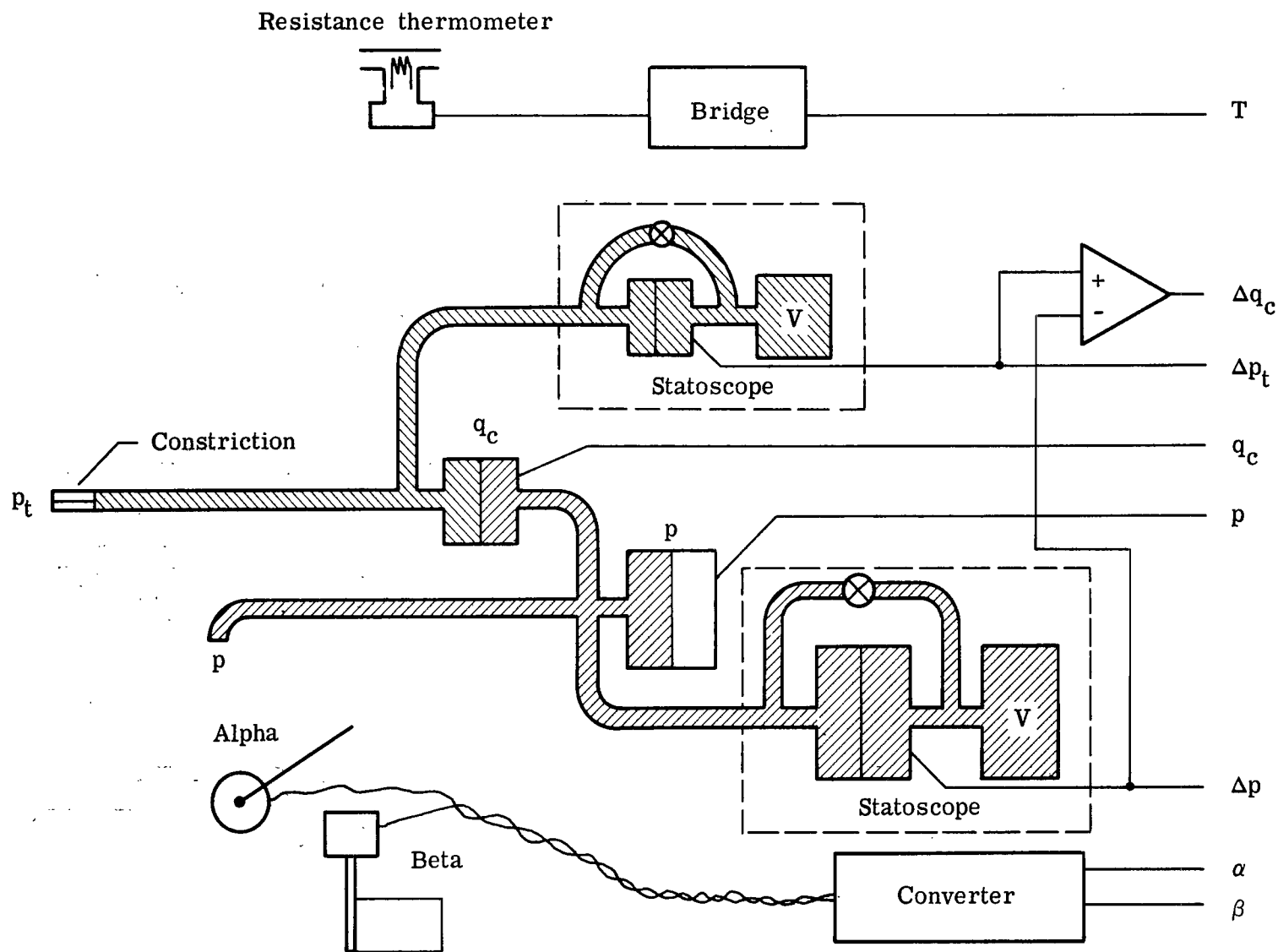


Figure 4.- Airflow measurement transducers.

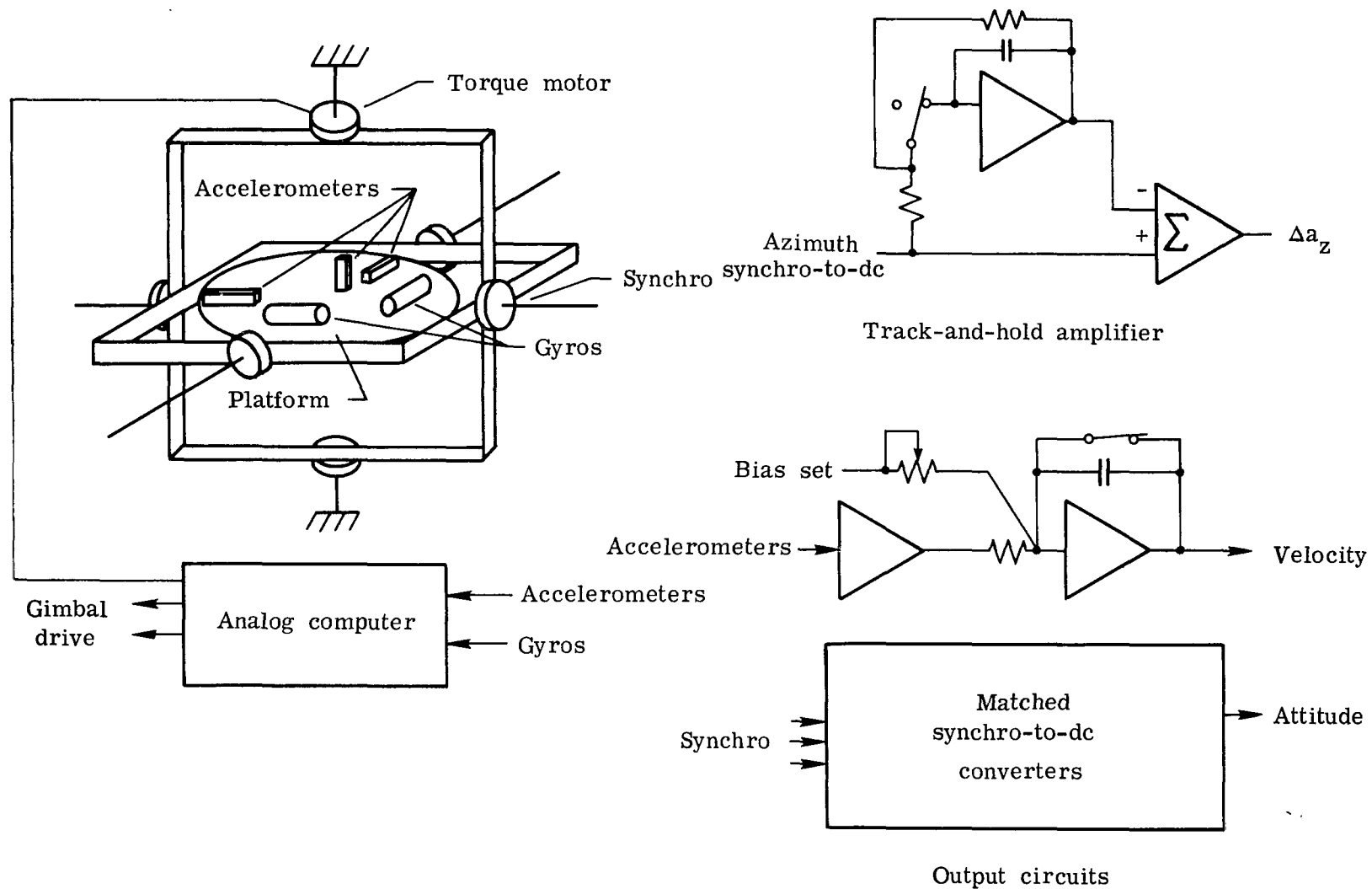


Figure 5.- Airplane response measurements.

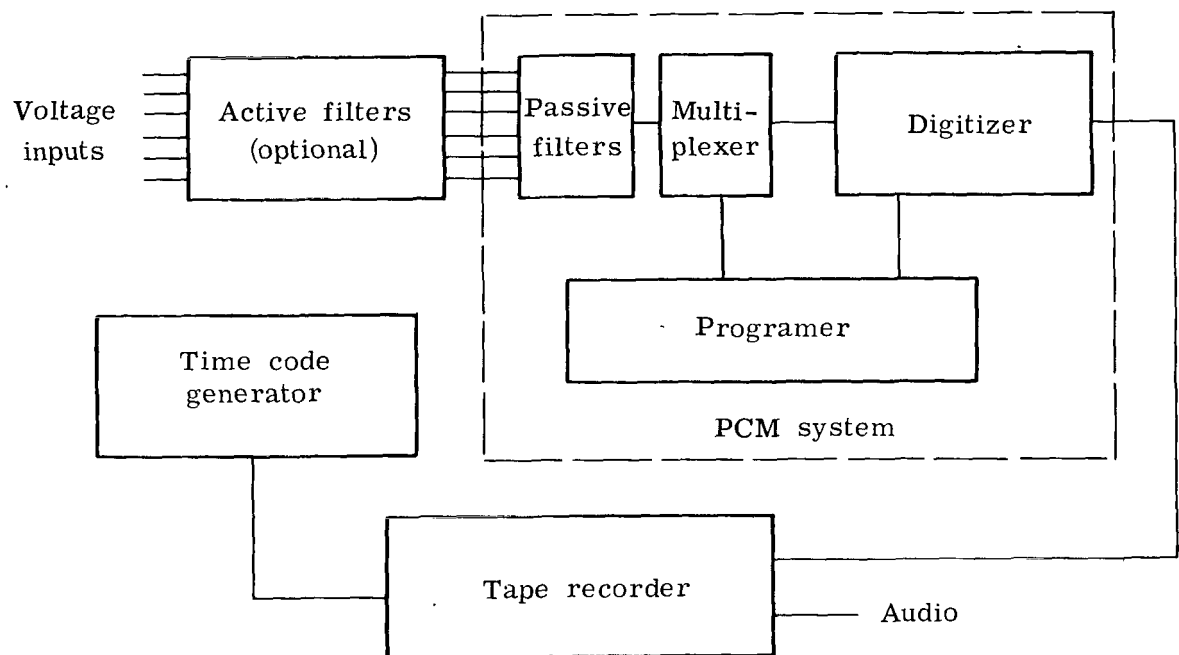


Figure 6.- Data system.

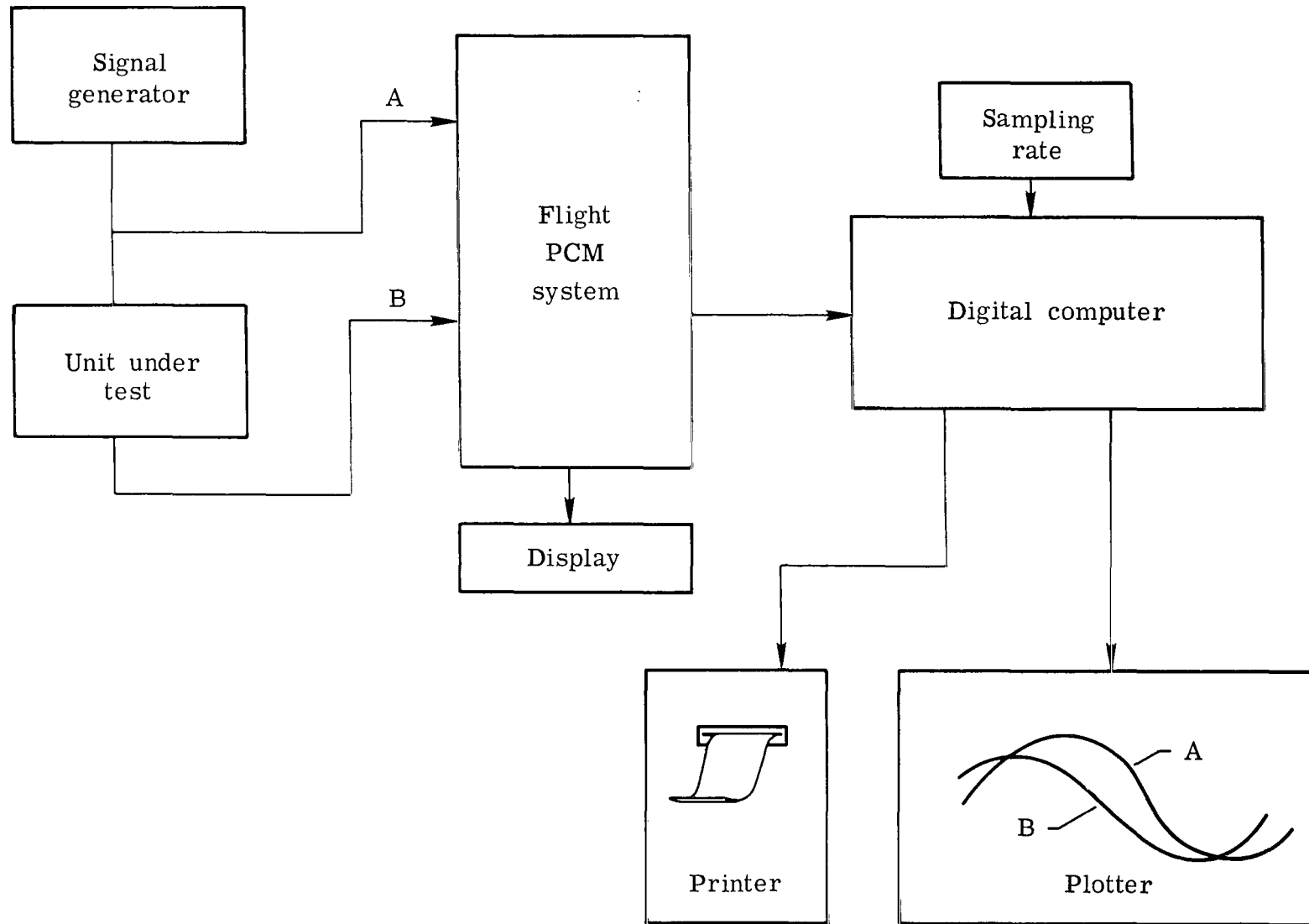


Figure 7.- Frequency-response measurement system.



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